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Journal of Mechanical Engineering

An International Journal

Volume 6 No. 1 April 2009 ISSN 1823-5514

JOURNAL OF MECHANICAL ENGINEERING (JMechE)

Journal of Mechanical Engineering is jointly published by the Faculty of Mechanical Engineering and University Publication Centre (UPENA), Universiti Teknologi MARA, 40450 Shah Alam, Selangor, Malaysia.

Journal of Mechanical Engineering

An International Journal

Volume 6 No. 1 **April 2009** **ISSN 1823-5514**

A Robust Adaptive Control System for High Performance Aircraft

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ABSTRACT

The design of a longitudinal controller to enhance the maneuverability of a high performance aircraft is described. Since such aircraft is often required to fly outside the linear aerodynamic regimes, the design is based on the nonlinear equations of motion. The controller is of adaptive type and it is robust with respect to unknown, but bounded uncertainties. The performance of the resulting controller is evaluated through simulations using hypothetical fighter aircraft parameters. The controller is demonstrated to be able to obtain good and consistent performances over a wide range of flight conditions.

Introduction

Modern high performance aircraft is often required to fly in the flight regimes where aerodynamic nonlinearities can no longer be neglected. Flight at high angles of attack is an example of such regimes. Often the capability to fly and perform maneuvers in such flight regimes is used as a measure of the superiority of a fighter. Conventional control designs, which are based on linearization of the dynamics of the aircraft at a representative operating point, are certainly not adequate to cope with the flight conditions, since the neglected nonlinearities become the determining factors affecting the dynamics of the aircraft. Better performances can be achieved by designing the flight control system based on the nonlinear model of aircraft dynamics [1].

The construction of an accurate model for the aircraft dynamics is a challenge. An aircraft can have several configurations depending on the mission, and thus the need for several dynamic representations. For example, the aerodynamic model of a fighter aircraft in its fully loaded configuration, with all its undercarriages attached on their ports, is different than in its clean configuration. Moreover, the environment where the aircraft is flying has been known to have significant effects on the dynamics. Therefore, a controller that is designed based on a particular condition may not perform well in other flight conditions and in the worst scenario, it may result in system instability. It is very desirable to have a flight control system that would yield an even performance on any point within the flight envelope.

The goal of the flight control system is in general to get a good if not optimum performance at any point within the flight envelope. Therefore, for a high performance aircraft with expanded flight envelope, the control design based on a specific operating condition of the aircraft is not adequate to achieve the goal. An approach that has been proposed to get a good performance in various flight condition is gain scheduling [2,3]. In this approach, the control gains are changed (usually discontinuously) depending on the operating condition of the aircraft. The design itself is usually based on the linearization of the equations of motion at several operating conditions within the flight envelope, hereby called the design operating conditions. The drawback of the approach is that there is no guarantee on its performance, or even worse, no guarantee on its stability when the aircraft flies outside its design operating conditions.

Nonlinear control approaches have also been attempted for flight control design. In [4,5], a variable structure controller approach (e.g. sliding controller) is used. Using such approach, sufficient stability condition can be easily guaranteed for the whole possible flight conditions. The resulting performances, however, can vary from one flight condition to another. Adaptive control methods have also been attempted, for example in [6,7]. In these works, however, the adaptive controllers are designed based on the linear dynamic models of the aircraft. Hence, the controller may not work well in the flight regimes where the nonlinearities come into play. An approach combining some of the methods mentioned above has also been reported, for example [8], where the adaptive technique is combined with the sliding mode method.

In [9], several different nonlinear controllers with application on complex aircraft control design are compared, including dynamic inversion, model predictive control, variable structure control, fuzzy logic, and adaptive type of controllers. Although each of the controllers considered has its strengths and weaknesses, there is no clear-cut winner. Therefore, the choice of methods to use in a control design is very much dependent on the problem under consideration.

In this paper, a controller that allows an aircraft to perform agile longitudinal maneuvers over a wide range of operating conditions is considered. The controller has to be able to control the aircraft to follow specific angle of attack and pitch angle commands. The capability to perform such maneuver is desirable, for example, to orient the nose of the aircraft to track a target while flying in certain direction for release of weapons. The design is based on the nonlinear equations of motion of the aircraft. An adaptive approach, which is guaranteed to be stable, is used to obtain good and consistent performances over a wide range of flight conditions. The modification on the adaptive controller to achieve robustness to unmodeled dynamics is also considered. Finally, the performance achieved using the controller is demonstrated through simulations.

Nonlinear Longitudinal Dynamics Model

In this paper, the aircraft is assumed rigid and its motion is restricted to the vertical plane only. The lateral-directional motions as well as the possible coupling with the longitudinal motion are neglected in this work. Thus, only the longitudinal equations of motion are relevant in describing the aircraft dynamics. The body-fixed reference frame, $Ox_b y_b z_b$, as shown in Figure 1 is used in expressing the equations of motion.

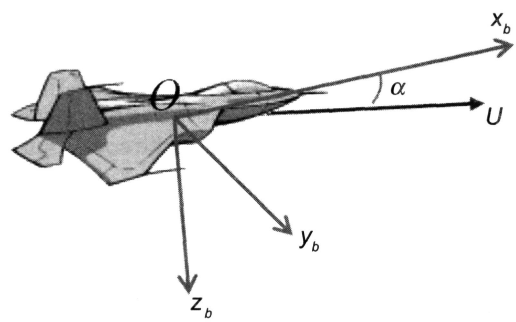

Figure 1: Aircraft Body-fixed Reference Frame.

Because of the maneuvers considered, the nominal forward airspeed (U) of the aircraft is assumed constant in this work, which in reality can be achieved by adjusting the thrust. For this reason, the thrust is no longer available as a control variable. For the longitudinal maneuvers, the aircraft is assumed to have elevators and flaperons for the controls. With these assumptions, the longitudinal dynamics can be described using three equations associated with angle of attack (α), pitch angle (θ), and pitch rate (q), as follows:

$$\begin{aligned}
\dot{\alpha} &= \frac{1}{1+\tan^2\alpha}\left[Uq + q\cos\theta - \left(L\cos\alpha + D\sin\alpha\right)/m\right] \\
\dot{q} &= \frac{M}{I_{yy}} \\
\dot{\theta} &= q
\end{aligned} \tag{1}$$

where m is the mass of the aircraft and I_{yy} is the aircraft moment of inertia about the y_b-axis. L, D, and M are the aerodynamic lift, drag, and pitching moment, respectively, which in this case are modeled according to the followings:

$$L = \bar{q}S\left[C_{l_0} + C_{l_\alpha}(\alpha)\alpha + C_{l_q}(\alpha)\left(\frac{\bar{c}}{2V_t}\right)q + C_{l_{\delta e}}\delta_e + C_{l_{\delta f}}\delta_f\right]$$

$$D = \bar{q}S\left[C_{d_0} + C_{d_\alpha}\alpha + C_{d_{\delta e}}\delta_e + C_{d_{\delta f}}\delta_f\right]$$
$$M = \bar{q}S\bar{c}\left[C_{m_0} + C_{m_\alpha}(\alpha)\alpha + C_{m_q}(\alpha)\left(\frac{\bar{c}}{2V_t}\right)q + C_{m_{\delta e}}\delta_e + C_{m_{\delta f}}\delta_f\right] \quad (2)$$

where $\bar{q}$ and V_t are the dynamic pressure and the total airspeed, respectively, defined below:

$$q = \tfrac{1}{2}\rho V_t^2$$
$$V_t = \sqrt{U^2 + w^2} = U\left(1 + \tan^2\alpha\right) \quad (3)$$

In the above equations, ρ is the air density, S is the wing planform area, and $\bar{c}$ is the mean aerodynamic chord. δ_e and δ_f are the elevator and flaperon deflection angles, respectively. In Equation (2), the coefficients C_x 's indicate the aircraft stability derivatives.

The control surfaces (elevators and flaperons) may have limitations in terms of their maximum deflections and rates. These limitations are not included in the controller design in this paper. However, the resulting control surface deflections will be used to evaluate the practicability of the controller. The deflections of the elevators and flaperons of more than ±30° are considered not realistic.

The stability derivatives are in general functions of angle of attack. For simplicity, however, only some are explicitly expressed as functions of angle of attack here. The dependency on angle of attack is in general influenced by the flight conditions, although the variations usually follow certain patterns. For example, C_l vs. α curve can usually be described quite well using a cubic polynomial in all flight conditions; however the coefficients of the polynomial to represent the curve accurately might vary with respect to the flight condition. Usually, the values of these coefficients can only be obtained accurately through flight tests. For this reason, a control design that relies on the values of these coefficients is not appropriate and this justifies the need for an adaptive type of controller.

The following approximations are assumed in this work:

$$C_{l_\alpha} = C_{l_{\alpha 1}} - C_{l_{\alpha 2}}\alpha^2$$
$$C_{l_q} = C_{l_{q1}} - C_{l_{q2}}\alpha$$
$$C_{m_\alpha} = C_{m_{\alpha 1}} - C_{m_{\alpha 2}}\alpha^2 \quad (4)$$
$$C_{m_q} = C_{m_{q1}} - C_{m_{q2}}\alpha^2$$

By using the above approximations and by defining:

$$\bar{q}^* = \tfrac{1}{2}\rho U^2$$
$$p_\alpha = 1 + \tan^2\alpha \quad (5)$$

the equations of motion (1) can be written as:

$$
\begin{aligned}
\dot{\alpha} &= \frac{1}{p_\alpha}\left(q+\frac{g}{U}\cos\theta\right)-\frac{\bar{q}*S}{mU}\left[\left(C_{l_0}+C_{l_{\alpha 1}}\alpha-C_{l_{\alpha 2}}\alpha^3+\left(C_{l_{q1}}-C_{l_{q2}}\alpha\right)\right.\right.\\
&\left.\left(\frac{\bar{c}}{2V_t}\right)q\right]\cos\alpha+\left(C_{d_0}+C_{d_\alpha}\alpha\right)\sin\alpha\Big]-\frac{\bar{q}*S}{mU}\left(C_{d_{\delta e}}\cos\alpha+C_{d_{\delta e}}\sin\alpha\right)\\
&\left.\delta_e+\left(C_{l_{\delta f}}\cos\alpha+C_{d_{\delta f}}\sin\alpha\right)\delta_f\right]\\
\dot{q} &= \frac{\bar{q}*S\bar{c}}{I_{yy}}\left[\left(C_{m_0}+C_{m_{\alpha 1}}\alpha-C_{m_{\alpha 2}}\alpha^3+\left(C_{m_{q1}}-C_{m_{q2}}\alpha\right)\left(\frac{\bar{c}}{2V_t}\right)q\right)\right]p_\alpha\\
&+\frac{\bar{q}*S\bar{c}}{I_{yy}}\left(C_{m_{\delta e}}\delta_e+C_{m_{\delta f}}\delta_f\right)p_\alpha\\
\dot{\theta} &= q
\end{aligned}
\tag{6}
$$

The controller will be designed based on this set of equations.

Adaptive Controller Design

Equation (5) can be written in the form that is more suitable for the controller design, as follows:

$$
\begin{aligned}
\dot{\mathbf{z}} &= \mathbf{h}-Y\mathbf{a}+\mathbf{v}\\
\dot{\theta} &= q
\end{aligned}
\tag{7}
$$

where

$$
\begin{aligned}
\mathbf{z} &= \{\alpha \quad q\}^T\\
\mathbf{h} &= \{(q+(g/U)\cos\theta)/p_\alpha \quad 0\}^T\\
\mathbf{a} &= \{\mathbf{a}_1^T \quad \mathbf{a}_2^T\}^T \text{ with}\\
\mathbf{a}_1 &= (\bar{q}^*S/m)\{C_{l_0} \quad C_{l_{\alpha 1}} \quad C_{l_{\alpha 2}} \quad C_{l_{q1}} \quad C_{l_{q2}} \quad C_{d_0} \quad C_{d_\alpha}\}^T\\
\mathbf{a}_2 &= (\bar{q}^*S\bar{c}/I_{yy})\{C_{m_0} \quad C_{m_{\alpha 1}} \quad C_{m_{\alpha 2}} \quad C_{m_{q1}} \quad C_{m_{q2}}\}^T
\end{aligned}
$$

$$\mathbf{v} = B\mathbf{u} \text{ with}$$

$$\mathbf{u} = \{\delta_e \quad \delta_f\}^T$$

$$B = \begin{bmatrix} -(\bar{q} * S/m)\left(C_{l_{\delta e}} \cos\alpha + C_{d_{\delta e}} \sin\alpha\right) p_\alpha & -(\bar{q} * S/mU)\left(C_{l\delta f} \cos\alpha + C_{d\delta f} \sin\alpha\right) \\ \left(\bar{q} * S\bar{c}/I_{yy}\right) C_{m_{\delta e}} p_\alpha & \left(\bar{q} * S\bar{c}/I_{yy}\right) C_{m_{\delta f}} p_\alpha \end{bmatrix}$$

$$Y = [Y_1 \quad Y_2] \text{ with}$$

$$Y_1 = \begin{bmatrix} \cos\alpha & \alpha\cos\alpha & -\alpha^3\cos\alpha & (\bar{c}/2V_t)q\cos\alpha & -(\bar{c}/2V_t)\alpha q\cos\alpha & \sin\alpha & \alpha\sin\alpha \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 \end{bmatrix}$$

$$Y_2 = \begin{bmatrix} 0 & 0 & 0 & 0 & 0 \\ -p_\alpha & -\alpha p_\alpha & \alpha^3 p_\alpha & -(\bar{c}/2V_t)qp_\alpha & (\bar{c}/2V_t)\alpha qp_\alpha \end{bmatrix} \tag{8}$$

The goal of the controller is to achieve good maneuver performances in following α and θ commands in any flight condition within the aircraft's flight envelope and within the limitations of the control surfaces. Such maneuver can be done in a relatively short period of time, and thus m and I_{yy} can be assumed constant during the maneuver. Hence for a specific flight condition, the vector a in Equation (7) consists of constant elements. The controller adaptation scheme will be applied to these constants in order to achieve good performances over various flight regimes. Further, measurements for airspeed, angle of attack, pitch angle, and pitch rate are assumed available for feedback. With this assumption, all the elements of the Y-matrix are known. The control derivatives are also assumed known, and therefore the elements of B-matrix are known. This last assumption is not too strict, however, since it is sufficient to know only their estimated values. The predicted errors of the estimation can then be considered as the unmodeled disturbances, which will be treated in the next section.

In Equation (6), **v** acts as the control variable. The adaptive controller will be designed based on **v**. After the appropriate **v** has been calculated, the actual elevator and flaperon responses can be found using

$$\mathbf{u} = B^{-1}\mathbf{v} \tag{9}$$

Since Equation (8) involves matrix inversion, the singularity condition needs to be examined. Matrix B will be singular when $\det(B) = 0$, which leads to the following singularity condition:

$$\tan\alpha = \frac{C_{l_{\delta e}} C_{m_{\delta f}} - C_{m_{\delta e}} C_{l_{\delta f}}}{C_{m_{\delta e}} C_{d_{\delta f}} - C_{d_{\delta e}} C_{m_{\delta f}}} \tag{10}$$

For the typical values of control derivative coefficients, this condition is met only at relatively high α values, which are usually associated with flight conditions that are outside the flight envelope. For example, for the aircraft parameters used in the simulation (see Appendix), the above condition is met at $\alpha = -89°$ for the flight condition 1. Hence, it can be safely assumed that Equation (8) always leads to a solution within the flight envelope of the aircraft.

For the control design, an intermediate variable s is defined, as follows:

$$\mathbf{s} = \left(\frac{d}{dt} + \lambda\right)\left\{\begin{matrix} \int^{t} \tilde{\alpha} d\tau \\ \tilde{\theta} \end{matrix}\right\} = \dot{\mathbf{x}} - \dot{\mathbf{x}}_{\mathbf{r}} \tag{11}$$

with $\tilde{\alpha} = \alpha - \alpha_d$ and $\tilde{\theta} = \theta - \theta_d$, where the subscript d indicates the desired value, and

$$\mathbf{x} = \left\{\begin{matrix} \int^{t} \alpha d\tau \\ \theta \end{matrix}\right\} \tag{12}$$

$$\dot{\mathbf{x}}_r = \dot{\mathbf{x}}_d - \lambda(\mathbf{x} - \mathbf{x}_d)$$

Note that in Equation (10), an integration constant is defined within the integral of $\tilde{\alpha}$. As will be shown later, this integration constant is useful to prevent the use of an excessive control effort, which may cause control saturation.

Continuing with the controller design, the Lyapunov-like function below is defined:

$$V = \tfrac{1}{2}\mathbf{s}^T\mathbf{s} + \tfrac{1}{2}\tilde{\mathbf{a}}^T\Gamma^{-1}\tilde{\mathbf{a}} \tag{13}$$

In the above equation, $\tilde{\mathbf{a}} = \hat{\mathbf{a}} - \mathbf{a}$, where $\hat{\mathbf{a}}$ indicates the estimate of $\mathbf{a}$. Γ is a constant symmetric positive definite matrix, and hence, V is a positive definite function. The time derivative of V is

$$\dot{V} = \mathbf{s}^T(\mathbf{h} - Y\mathbf{a} + \mathbf{v} - \ddot{\mathbf{x}}_{\mathbf{r}}) + \dot{\hat{\mathbf{a}}}^T\Gamma^{-1}\tilde{\mathbf{a}} \tag{14}$$

Based on Equation (13), the following control law is selected:

$$\mathbf{v} = \ddot{\mathbf{x}}_{\mathbf{r}} - \mathbf{h} + Y\hat{\mathbf{a}} - K_d\mathbf{s} \tag{15}$$

where K_d is a constant symmetric positive definite matrix. By using this control law, Equation (13) becomes

$$\dot{V} = \mathbf{s}^T(Y\tilde{\mathbf{a}} - K_d\mathbf{s}) + \dot{\hat{\mathbf{a}}}^T\Gamma^{-1}\tilde{\mathbf{a}} \tag{16}$$

Then, by choosing the adaptation law:

$$\dot{\mathbf{a}} = -\Gamma Y^T \mathbf{s} \tag{17}$$

Equation (15) becomes

$$\dot{V} = -\mathbf{s}^T K_d \mathbf{s} \tag{18}$$

which is negative definite. It can be shown that $\ddot{V}$ is bounded, so that by Barbalat's lemma [10], $\dot{V} \rightarrow 0$ as $t \rightarrow \infty$, which implies that $\mathbf{s} \rightarrow 0$ as $t \rightarrow \infty$. In other words, the tracking errors converge to zero in steady state.

It can be observed from Equation (14) that the magnitude of the control depends on the magnitude of **s**. If **s** is large, the control action will also be large. If a step command is given to the system, then initially **s** is relatively large. To prevent the use of an excessive control effort, the integration constant in **s** can be adjusted such that the magnitude of **s** is small initially.

Modification for Robustness

The adaptive controller obtained in the previous section is modified further here to cope with the unmodeled disturbances, which could be contributed by modeling errors and time-dependent disturbances experienced by the aircraft, such as due to gusts or turbulences. Often times, although such disturbances are hard to model, their magnitudes can usually be predicted. It is assumed here that the bounds on the magnitude of such disturbances are known. The equations of motion of the aircraft with disturbances are

$$\begin{aligned} \dot{\mathbf{z}} &= \mathbf{h} - Y\mathbf{a} + \mathbf{v} + \mathbf{d} \\ \dot{\theta} &= q \end{aligned} \tag{19}$$

where **d** is the unmodeled disturbance vector, which in general is a function of state variables and time. It is assumed

$$|d_i| \leq D_i \quad ; \quad i = 1,2 \tag{20}$$

with D_i known.

The intermediate variables (10) are modified as follows:

$$s_{\Delta i} = s_i - \phi_i \operatorname{sat}\left(\frac{s_i}{\phi_i}\right); i = 1,2 \tag{21}$$

where ϕ_i with i=1,2 are positive constants to be determined such that the stability of the system can be guaranteed in the presence of the disturbances. It is clear that:

$$\dot{\mathbf{s}}_{\Delta} = \dot{\mathbf{s}} \tag{22}$$

In the Lyapunov-like function (12), $\mathbf{s}_{\Delta}$ will be used instead of $\mathbf{s}$, which yields:

$$V = \tfrac{1}{2}\mathbf{s}_{\Delta}^{T}\mathbf{s}_{\Delta} + \tfrac{1}{2}\tilde{\mathbf{a}}^{T}\Gamma^{-1}\tilde{\mathbf{a}} \tag{23}$$

In this case, the same control law as given in Equation (14) is selected, so that

$$\dot{V} = \mathbf{s}_{\Delta}^{T}\left(Y\tilde{\mathbf{a}} - K_{d}\mathbf{s} + \mathbf{d}\right) + \dot{\hat{\mathbf{a}}}^{T}\Gamma^{-1}\tilde{\mathbf{a}} \tag{24}$$

Then by choosing the adaptation law:

$$\dot{\hat{\mathbf{a}}} = -\Gamma Y^{T}\mathbf{s}_{\Delta} \tag{25}$$

$\dot{V}$ becomes:

$$\dot{V} = -\mathbf{s}_{\Delta}^{T}K_{d}\mathbf{s}_{\Delta} + \mathbf{s}_{\Delta}^{T}\left(\mathbf{d} - K_{d}\begin{bmatrix}\phi_{1}\mathrm{sat}(s_{1}/\phi_{1})\\ \phi_{2}\mathrm{sat}(s_{2}/\phi_{2})\end{bmatrix}\right) \tag{26}$$

The second term in Equation (25) has the following properties:

$$\mathbf{s}_{\Delta}^{T}\left(\mathbf{d} - K_{d}\begin{bmatrix}\phi_{1}\mathrm{sat}(s_{1}/\phi_{1})\\ \phi_{2}\mathrm{sat}(s_{2}/\phi_{2})\end{bmatrix}\right) = \begin{cases} 0 \;\; ; \;\; \text{if } \mathbf{s}_{\Delta} = \mathbf{0} \\ \mathbf{s}_{\Delta}^{T}\left(\mathbf{d} - K_{d}\begin{bmatrix}\phi_{1}\mathrm{sgn}(s_{1}/\phi_{1})\\ \phi_{2}\mathrm{sgn}(s_{2}/\phi_{2})\end{bmatrix}\right) \;\; ; \;\; \text{if } \mathbf{s}_{\Delta} \neq \mathbf{0} \end{cases} \tag{27}$$

Therefore, to guarantee the stability of the system, the following condition has to be met:

$$\mathbf{s}_{\Delta}^{T}\left(\mathbf{d} - K_{d}\begin{bmatrix}\phi_{1}\mathrm{sgn}(s_{1}/\phi_{1})\\ \phi_{2}\mathrm{sgn}(s_{2}/\phi_{2})\end{bmatrix}\right) \leq 0 \tag{28}$$

This can be achieved by selecting ϕ_i with $i=1,2$ such that the inequality (28) is satisfied at any time. In the simplest case when K_d is diagonal, a simple choice of ϕ_i that satisfies the inequality (28) is

$$\phi_{i} = \frac{D_{i}}{K_{d_{i}}} \quad ; \quad i = 1,2 \tag{29}$$

When Equation (28) is satisfied, it can be shown that $\ddot{V}$ is bounded, and thus, by Barbalat's lemma, $\mathbf{s}_{\Delta} \to 0$ as $t \to \infty$ [10]. This implies that $\mathbf{s}$ will converge to the region where $|s_i| \leq \phi_i\,; i = 1,2$. In this case, the tracking errors do not converge to zero, however after some time their magnitudes are bounded by $|\phi_i|$.

Since ϕ_i is at our disposal, $|\phi_i|$ can be made small by selecting relatively large K_{d_i}.

Note that the adaptation law (24) depends on $\mathbf{s}_\Delta$. Hence, no adaptation is done in the region where $|s_i| \leq \phi_i\ ; i = 1,2$. Essentially, for robustness to unmodeled dynamics, an adaptation deadzone is given to the system. This adaptation deadzone is introduced to prevent the adaptation process from taking disturbances as useful information.

Simulations

The robust adaptive controller discussed previously is evaluated by applying it to a generic fighter aircraft. The longitudinal aircraft data for several flight conditions are given in Appendix. The data are typical for a small fixed-wing fighter aircraft. The disturbances used in the simulations are also given in Appendix. Unless specifically mentioned, the following values are used in the simulations:

$$\begin{aligned} &\Gamma = 10 I_{12}, \text{ where } I_n \text{ is an } n \text{ by } n \text{ identity matrix.} \\ &K_d = 5 I_2 \\ &\lambda = 4 \\ &D = [0.1 \quad 0.1]^T \end{aligned} \tag{29}$$

First, the effect of the integration constant in (10) is simulated. Figure 2 compares the responses of the aircraft in flight condition 1 (see Appendix) for two cases:

a. Zero integration constant, and

b. Integration constant $= -\dfrac{\theta_d}{\lambda}$, where θ_d is a step command of θ.

The choice of the integration constant value in case b is to make $s_1\,(t = 0) = 0$. As can be seen from the figure, the trajectories of s_1 differ significantly between the two cases. In case b, the magnitude of s_1 is kept small initially, which results in reduced control efforts when compared to case a. The reduction is especially significant for the flaperon, since it is directly influenced by s_1. This in effect reduces the rise time of the angle-of-attack response, but eliminates the overshoot in the response and it does not affect the time to reach the steady state. The effect of the integration constant on pitch angle response is only minor. For practical purposes, the reduction in the control efforts in case b. does not degrade the system performance, and in fact the response can be considered better due to the elimination of the overshoot in the angle-of-attack response. It can also be

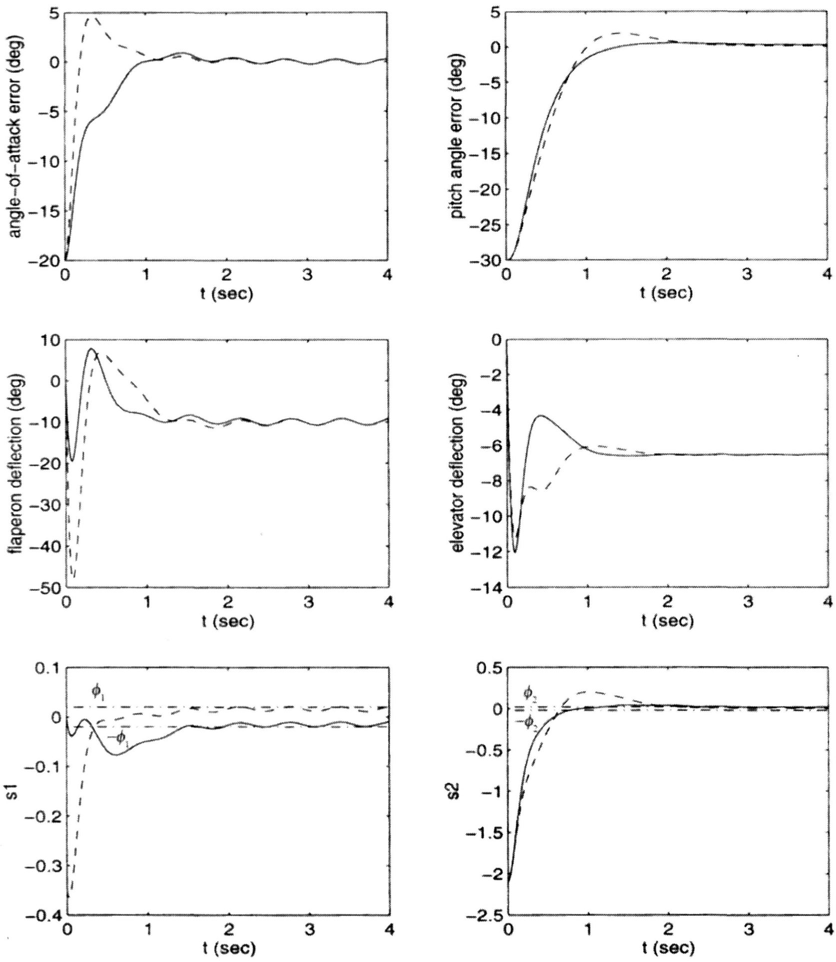

Figure 2: System responses using controller with zero integration constant (dashed curves) and with integration constant = $-\theta_d/\lambda$ (solid curves) for flight condition 1

observed from Figure 2 that the trajectories of s_1 and s_2 converge to the region where $|s_i| \le \phi_i\ ; i = 1,2$; as predicted by the theory.

Figure 3 shows the responses of the system for two different values of K_d and Γ.

The performance of the system for the larger K_d and Γ values is slightly better, however it also necessitates larger control efforts. In practice, a tradeoff should be made between the performance gain and the control effort for

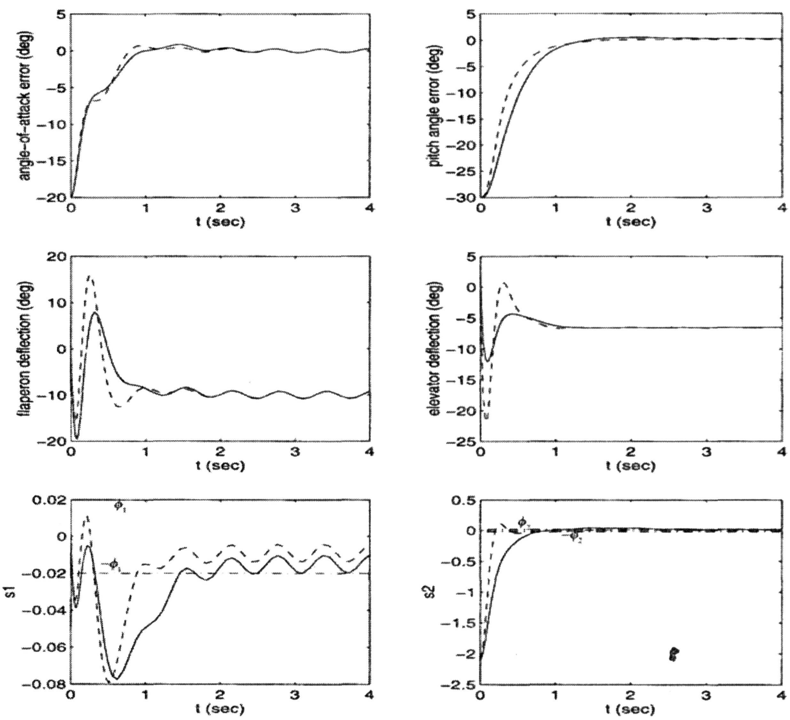

Figure 3: Comparison of responses using $\Gamma = 20I_{12}, K_d = \begin{bmatrix} 5 & 0 \\ 0 & 10 \end{bmatrix}$ (dashed curves) and using $\Gamma = 10I_{12}, K_d = 5I_2$ (solid curves) for flight condition 1

various conditions. K_d and Γ should be adjusted based on the result of such tradeoff.

The responses of the aircraft to the step commands of $\alpha_d = 20°$ and $\theta_d = 30°$ for four different flight conditions (as given in Appendix) are also simulated. In the simulations, the integration constant of $-\theta_d / \lambda$ is used. The

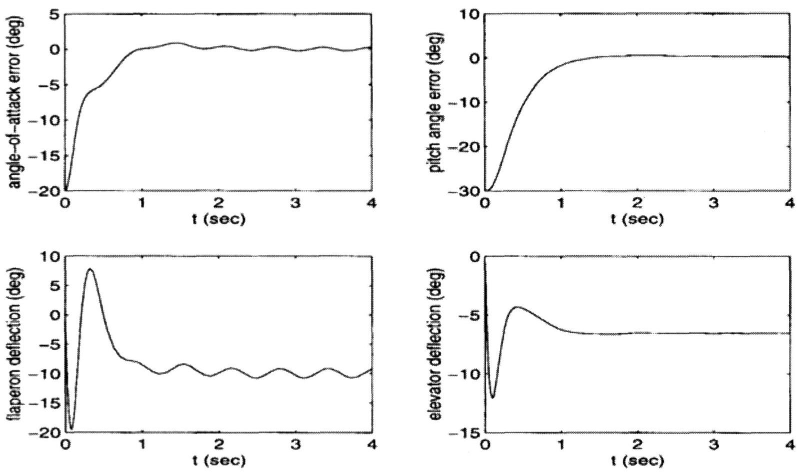

Figure 4: Responses to $\alpha_d = 20°$ and $\theta_d = 30°$ in Flight Condition 1

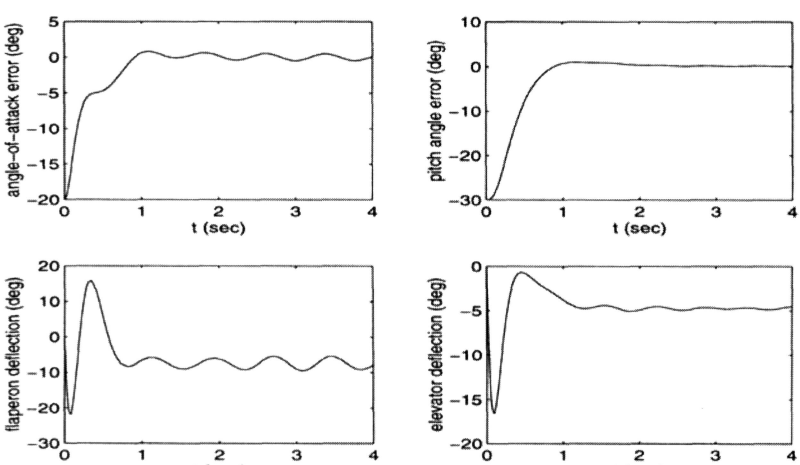

Figure 5: Responses to $\alpha_d = 20°$ and $\theta_d = 30°$ in Flight Condition 2

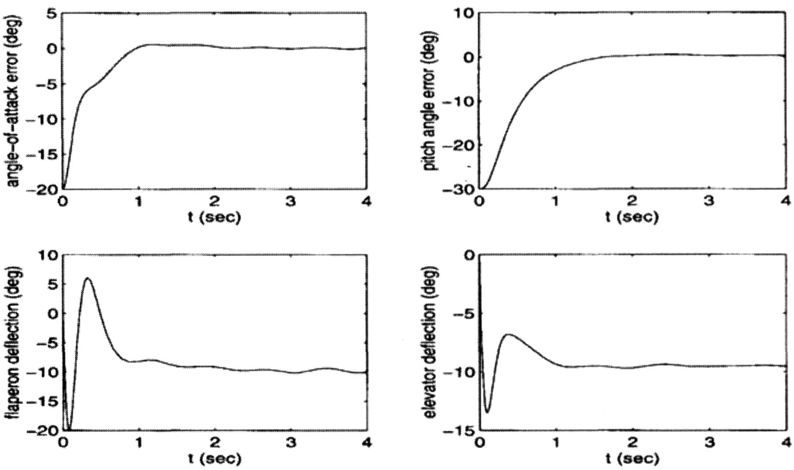

Figure 6: Responses to $\alpha_d = 20°$ and $\theta_d = 30°$ in Flight Condition 3

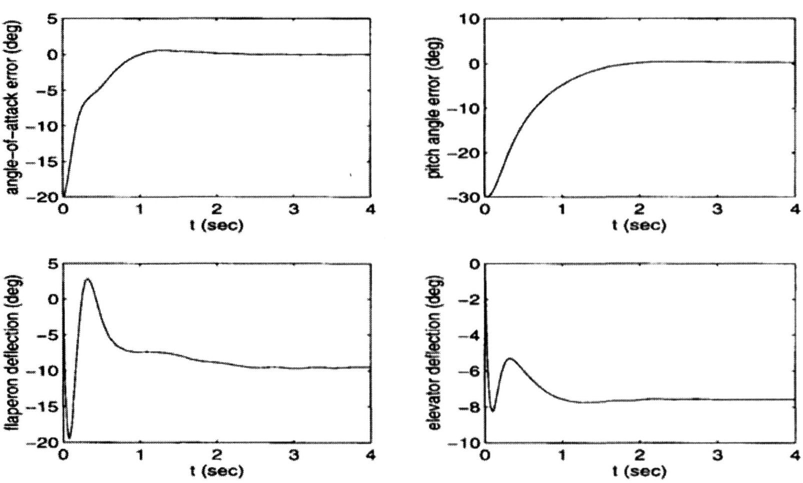

Figure 7: Responses to $\alpha_d = 20°$ and $\theta_d = 30°$ in Flight Condition 4

results of the simulations for zero initial values of the state variables and the parameter estimates are presented in Figures 4 to 7.

These results show that the controller designed using the robust adaptive approach performs fairly well, yielding consistent performances in different flight

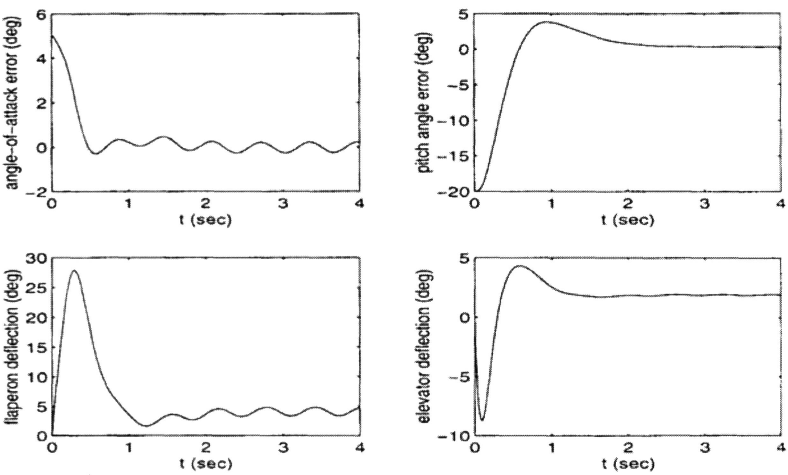

Figure 8: Responses to $\alpha_d = -5°$ and $\theta_d = 20°$ in Flight Condition 1

conditions. Some unnatural maneuvers are also simulated. Figure 8 shows an example of such a maneuver.

It can be observed that relatively large control efforts are needed for such a maneuver. Providing the necessary control actions are available, the controller also yields fairly good aircraft responses in such situation.

Conclusions

A longitudinal control system for a high performance aircraft has been designed using a robust adaptive technique. The resulting controller is guaranteed to be stable for a set of specified disturbance bounds. Reduction in control efforts with relatively no degradation in the resulting performance has also been discussed by adjusting the integration constant in the integral control part within the controller. The simulation shows that the controller can achieve good and consistent performances over a wide range of flight conditions.

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Appendix

Longitudinal aircraft data at various flight conditions

Flight Condition 1

$m = 9773$ kg, $I_{yy} = 127512$ kg m^2, $S = 33.75$ m^2
altitude = 0 ft, $M = 0.6$

$C_{l_0} = 0$	$C_{d_0} = 0$	$C_{m_0} = 0$
$C_{l_{\alpha 1}} = 4$	$C_{d_\alpha} = 0.45$	$C_{m_{\alpha 1}} = -0.6$
$C_{l_{\alpha 2}} = 6$	$C_{d_{\delta e}} = 0.01$	$C_{m_{\alpha 2}} = -0.8$
$C_{l_{q1}} = 2.5$	$C_{d_{\delta f}} = 0.1$	$C_{m_{q1}} = -2$
$C_{l_{q2}} = 0$		$C_{m_{q2}} = 12$
$C_{l_{\delta e}} = 0.68$		$C_{m_{\delta e}} = -1.46$
$C_{l_{\delta f}} = 4$		$C_{m_{\delta f}} = -0.05$

Disturbance: $d_1 = 0.04 \sin 10t$ $\quad d_2 = 0.06 \sin 8t$

Flight Condition 2

$m = 9500$ kg, $I_{yy} = 121000$ kg m^2, $S = 33.75$ m^2
altitude = 5000 ft, $M = 0.6$

$C_{l_0} = 0$	$C_{d_0} = 0.04$	$C_{m_0} = 0$
$C_{l_{\alpha 1}} = 4$	$C_{d_\alpha} = 0.4$	$C_{m_{\alpha 1}} = -0.4$
$C_{l_{\alpha 2}} = 12$	$C_{d_{\delta e}} = 0.01$	$C_{m_{\alpha 2}} = -0.8$
$C_{l_{q1}} = 2$	$C_{d_{\delta f}} = 0.1$	$C_{m_{q1}} = 0$
$C_{l_{q2}} = 0$		$C_{m_{q2}} = 8$
$C_{l_{\delta e}} = 0.72$		$C_{m_{\delta e}} = -1.2$
$C_{l_{\delta f}} = 4$		$C_{m_{\delta f}} = -0.05$

Disturbance: $d_1 = 0.06 \sin 8t$ $\quad d_2 = 0.1 \sin 10t$

Flight Condition 3
$m = 9700$ kg, $I_{yy} = 123125$ kg m^2, $S = 33.75$ m^2
altitude = 9000 ft, $M = 0.8$

$C_{l_0} = 0$ $\quad C_{d_0} = 0.04$ $\quad C_{m_0} = 0$
$C_{l_{\alpha 1}} = 4$ $\quad C_{d_\alpha} = 0.45$ $\quad C_{m_{\alpha 1}} = -0.6$
$C_{l_{\alpha 2}} = 6$ $\quad C_{d_{\delta e}} = 0.01$ $\quad C_{m_{\alpha 2}} = -0.9$
$C_{l_{q1}} = 3$ $\quad C_{d_{\delta f}} = 0.1$ $\quad C_{m_{q1}} = -2$
$C_{l_{q2}} = 0$ $\quad C_{m_{q2}} = 10$
$C_{l_{\delta e}} = 0.68$ $\quad C_{m_{\delta e}} = -0.98$
$C_{l_{\delta f}} = 4$ $\quad C_{m_{\delta f}} = -0.05$

Disturbance: $d_1 = 0.02\sin 8t$ $\qquad d_2 = 0.05\sin 6t$

Flight Condition 4
$m = 9840$ kg, $I_{yy} = 128000$ kg m^2, $S = 33.75$ m^2
altitude = 10000 ft, $M = 0.9$

$C_{l_0} = 0$ $\quad C_{d_0} = 0.04$ $\quad C_{m_0} = 0$
$C_{l_{\alpha 1}} = 3.9$ $\quad C_{d_\alpha} = 0.4$ $\quad C_{m_{\alpha 1}} = -0.65$
$C_{l_{\alpha 2}} = 7$ $\quad C_{d_{\delta e}} = 0.01$ $\quad C_{m_{\alpha 2}} = -0.9$
$C_{l_{q1}} = 3.5$ $\quad C_{d_{\delta f}} = 0.1$ $\quad C_{m_{q1}} = -1$
$C_{l_{q2}} = 4$ $\quad C_{m_{q2}} = 12$
$C_{l_{\delta e}} = 0.5$ $\quad C_{m_{\delta e}} = -1.36$
$C_{l_{\delta f}} = 4$ $\quad C_{m_{\delta f}} = -0.05$

Disturbance: $d_1 = 0.01\sin 12t$ $\qquad d_2 = 0.03\sin 10t$

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References

Use squared brackets to indicate reference citation such as [1], [3]-[5] in the main text. Include references at the end of the paper according to the citations order that appears in the paper using the following format.

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